

## REPORT No. 781

# WIND-TUNNEL PROCEDURE FOR DETERMINATION OF CRITICAL STABILITY AND CONTROL CHARACTERISTICS OF AIRPLANES

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### SUMMARY

*This report outlines the flight conditions that are usually critical in determining the design of components of an airplane which affect its stability and control characteristics. The wind-tunnel tests necessary to determine the pertinent data for these conditions are indicated, and the methods of computation used to translate these data into characteristics which define the flying qualities of the airplane are illustrated.*

### INTRODUCTION

The development of flying-qualities specifications (references 1, 2, and 3) has established specific criteria with which the characteristics of an airplane normally will be compared. The problem posed in the preliminary design of an airplane is the determination of which of these criteria will influence the design of the various components of the airplane that affect the stability and control characteristics, and the magnitude of the effect. As an aid in this design problem, methods have been developed by which the data, obtained from wind-tunnel tests of powered models, can be translated into flying-qualities characteristics observable in flight tests (in the terms of which the flying-qualities specifications are written). Application of these methods to six different airplanes has indicated that the same requirements represent the critical conditions on all conventional airplanes, and that if these conditions are met it will follow that the remainder of the specifications will be satisfied. By permitting concentration on these few conditions a considerable simplification of the design process results.

It is the purpose of this report to outline the critical conditions for each component of the airplane, to indicate the wind-tunnel tests necessary to determine the pertinent data, and to illustrate the methods of computation used to translate these data into characteristics which define the flying qualities of an airplane.

### DISCUSSION

The flying-qualities requirements can be stated under three major headings:

1. Stability shall exist under specified conditions.
2. Control shall exist under specified conditions.
3. Control forces shall be kept within specified limits.

Each of these requirements is, to some extent, contradictory to the other two and, furthermore, airplanes now have been developed to such sizes and powers that the attainment of all three requirements is quite difficult. Hence, despite the

fact that from the ultimate flying-qualities standpoint it is desirable to satisfy some of the requirements by as ample a margin as possible, the designer normally will find it expedient to base his original design on small margins, in order to minimize the difficulty of compromising conflicting requirements. If this is not done for one requirement, the attainment of the other two by normal means may be impossible.

To illustrate this point, the horizontal tail on a typical high-powered, single-engine airplane must be the smallest which will give the required stability in a rated-power climb, and the elevator must be the smallest which will give the required control in landing, in order to keep the balance requirements for low control forces in accelerated maneuvers within reasonable limits. With regard to wing dihedral, care must be taken not to exceed the amount required for the maintenance of lateral stability in the low-speed, high-power condition where the dihedral effect will be minimum, or excessive dihedral effect will result at high speeds. The size of the rudder must be limited to the smallest that will give adequate control in order to keep the rudder-pedal forces within the required limits.

If it is assumed that the preliminary design has been completed on the above basis, it will be the function of the first wind-tunnel tests to obtain data from which any readjustments in the airplane components, necessary to secure satisfactory characteristics, can be determined. As conceived herein, the first series of wind-tunnel tests would be restricted to the critical conditions with regard to each characteristic. A series of tests sufficiently complete to form a basis for a more general flying-qualities prediction, or an analysis of secondary effects, would not be made until the changes shown to be necessary by the first series of tests had been incorporated in the model. An outline for such a preliminary series of tests as just discussed is given in tables I, II, and III for a single-engine airplane and in tables I, II, and IV for a twin-engine airplane. An attempt has been made to make these tables self-explanatory when considered in the light of a flying-qualities specification (references 1, 2, and 3). Figures 1 to 16 present a typical set of results. The method of translating the wind-tunnel results into the terms of the flying-qualities specification is outlined in these figures.

The choice of critical conditions and the tables have been made after a detailed study of the characteristics of three typical single-engine airplanes and three twin-engine airplanes, with right-hand rotating propellers. In each case,

it was found that if the 10 major points as outlined were satisfied, the other characteristics called for in the flying-qualities specifications would be met. It is believed that this conclusion will be similar for other conventional airplanes.

Each of the ten items listed in the tables is directed toward one major variable in the airplane design. Thus, in the usual case

*Horizontal tail size* will be determined by item I.

*Elevator size* will be determined by item II.

*Elevator balance* will be determined by item III.

*Minimum dihedral* will be determined by item IV.

*Maximum dihedral* will be determined by item V.

*Aileron size* will be determined by item VI.

*Aileron balance* will be determined by item VII.

*Vertical tail size* will be determined by item VIII.

*Rudder size* will be determined by item IX.

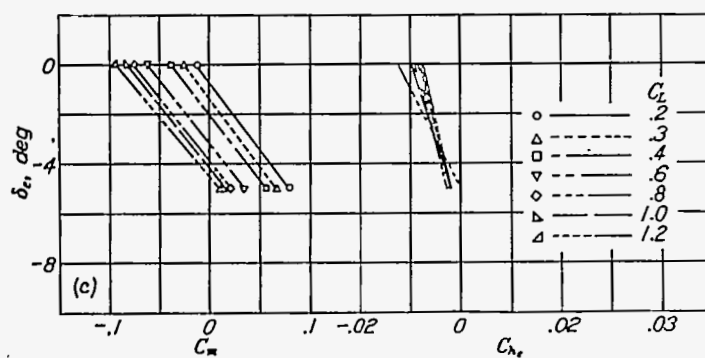
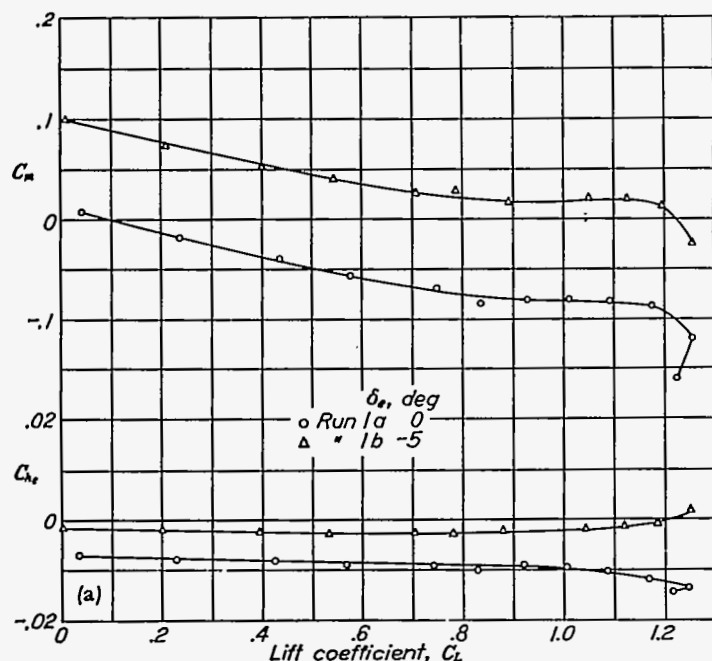
*Rudder balance* will be determined by item X.

Obviously there is a closer interrelation among the characteristics than the above listing implies, and important changes can be required after consideration of "secondary" variables. However, to a first approximation the variables listed will establish the airplane stability and control characteristics after the first basic arrangement of wing and fuselage is established. Changes in other features of the airplane components will normally be in the nature of refinements, rather than major changes.

The surface deflections given in the text are only representative values corresponding to the range of deflections needed in ascertaining the flying qualities of the airplanes upon which the study has been based. An optimum selection can be best determined from a cursory examination of the basic runs with control surfaces neutral, with due regard for the maximum deflections upon which the design is based. It will be noted that tail-off runs are called for in the tables only when they are necessary for the computation of the flying qualities. However, in order to provide data which will aid in any necessary redesign, the addition of a tail-off run for other test conditions is considered desirable.

A typical set of data as obtained from the runs called for on the tables is shown in the figures and the cross plots and computation methods necessary to reduce these data to the form of the flying-qualities characteristics are outlined. As in the table, these figures are intended to be in such detail as to require no further explanation. In the computation procedure certain simplifications and assumptions have been made, but it is believed that all factors which will bear an important influence on the final result have been included.

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①	②	③	④	⑤	⑥	⑦	⑧
$C_L$	$V_i$ , m.p.h.	Neutral tab			Tab $C_{L_1}$ with tab set for trim at best climb speed	$C_{L_1}$ with trim at best climb speed	Stick force from $\frac{F}{qC_{L_1}} = 23.8$
		$\delta_e$ for $C_m$ equals zero from cross-plot	$C_{L_1}$ for ④ from cross-plot	$C_{L_1}$ for ⑤ at best climb speed ( $C_L=0.4$ )			
0.2	252	-0.6° (up)	-0.0065	-0.0057	+0.0057 × (1)	-0.0008	3.1 (push)
0.3	206	-1.3° (up)	-0.0063	-----	+0.0057 × (1)	-0.0006	1.6 (push)
0.4	178	-1.9° (up)	-0.0057	-----	+0.0057 × (1)	0	0
0.6	146	-3.1° (up)	-0.0047	-----	+0.0057 × (1.04)	+0.0012	1.6 (pull)
0.8	126	-4.0° (up)	-0.0032	-----	+0.0057 × (1.10)	+0.0031	3.0 (pull)
1.0	113	-4.2° (up)	-0.0027	-----	+0.0057 × (1.19)	+0.0041	3.2 (pull)
1.2	103	-4.4° (up)	-0.0006	-----	+0.0057 × (1.32)	+0.0069	4.9 (pull)

<sup>1</sup> The variation in  $C_{L_1}$  in column ⑥ is due to the variation of  $q/dq$  at the tail. The factor involved, given in column ⑥, may be obtained from the ratios of  $dC_m/d\delta_e$  in figures 1(c) and 4(c) with due regard for the relative location of the tab and slipstream. Wing loading = 32.6 lb./sq. ft.

(b) Computation table.

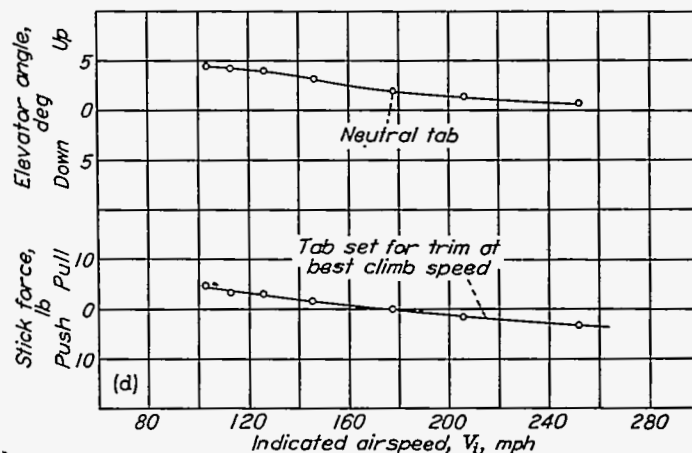
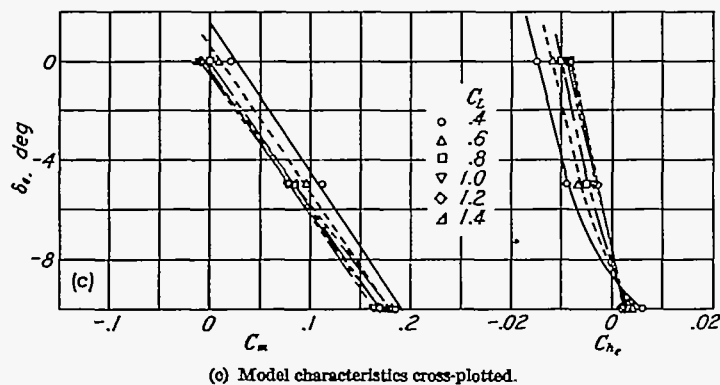
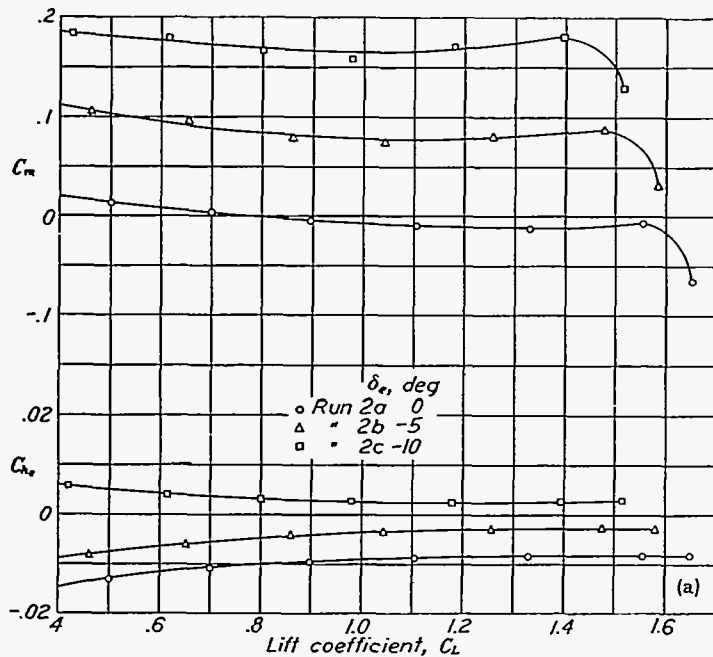


FIGURE 1.—Variation of elevator angle and stick force with speed. Steady flight with flaps and gear up and rated power. Single-engine airplane.



①	②	③	④	⑤	⑥	⑦	⑧
$C_L$	$V_i$ m.p.h.	Neutral tab			Tab $C_{h_e}$ with tab set for trim at $1.4 V_{stall}$	$C_{h_e}$ with trim at $1.4 V_{stall}$	Stick force from $\frac{F}{q C_{h_e}} = 23.8$
		$\delta_e$ for $C_m$ equals zero from cross plot	$C_{h_e}$ for ③ from cross plot	$C_{h_e}$ for ⑤ at $1.4 V_{stall}$ ( $C_L = 0.92$ ) from cross plot			
0.4	178	1.5°	-0.0170	-0.0095	0.0095	-0.0075	14.5 (push)
0.6	146	1.5°	-0.0130	-----	.0095	-.0035	4.5 (push)
0.8	126	0°	-0.0102	-----	.0095	-.0007	.7 (push)
1.0	113	-4°	-0.0083	-----	.0095	+.0007	.5 (pull)
1.2	103	-6°	-0.0080	-----	.0095	+.0015	1.0 (pull)
1.4	94	-6°	-0.0076	-----	.0095	+.0019	1.1 (pull)

<sup>1</sup> No allowance was made for  $q/q$  at the tail with respect to tab  $C_{h_e}$ . If necessary it can be done by the method in figure 1 (b).  
Wing loading = 32.6 lb./sq. ft.

(b) Computation table.

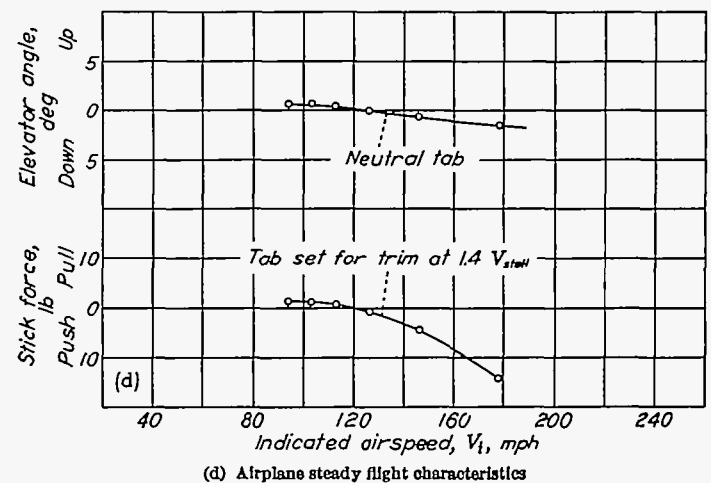
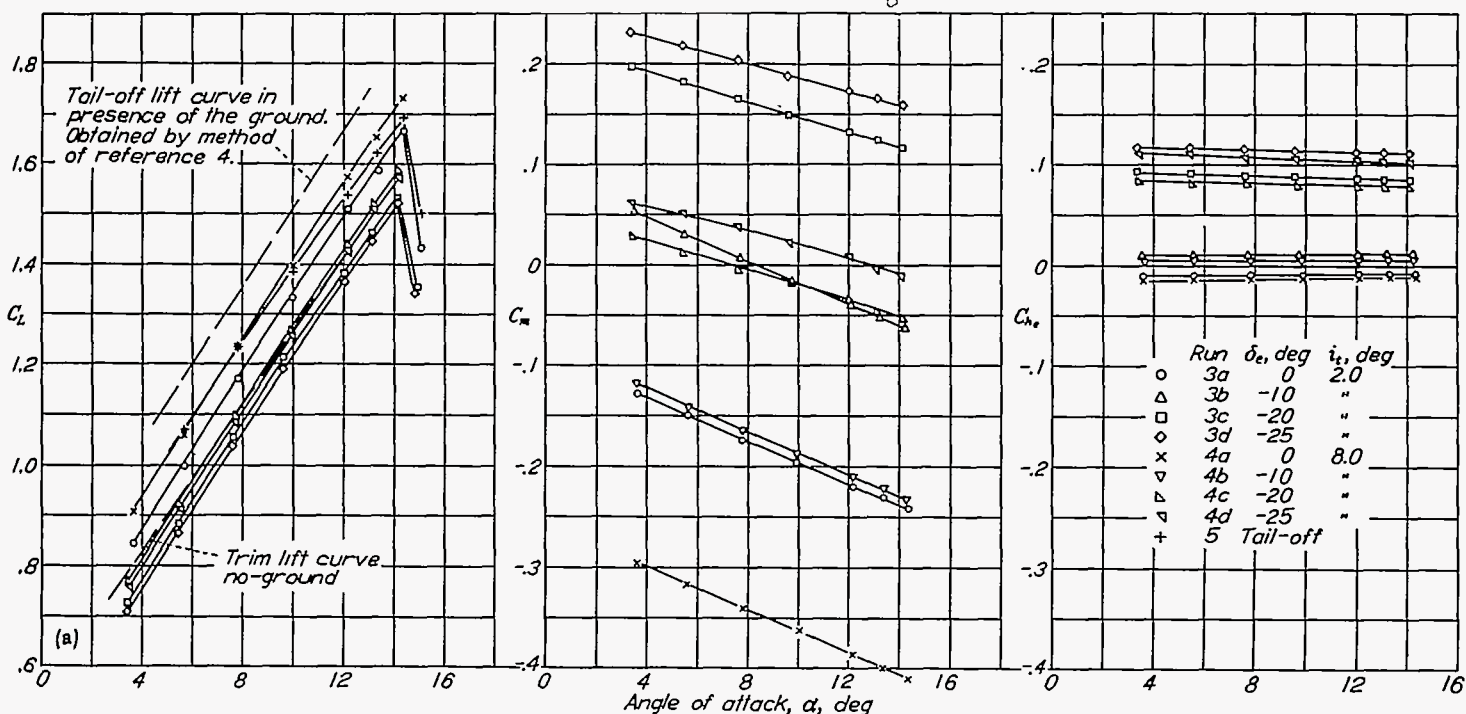


FIGURE 2.—Variation of elevator angle and stick force with speed. Steady flight with flaps and gear down, 50-percent normal rated power. Single-engine airplane.

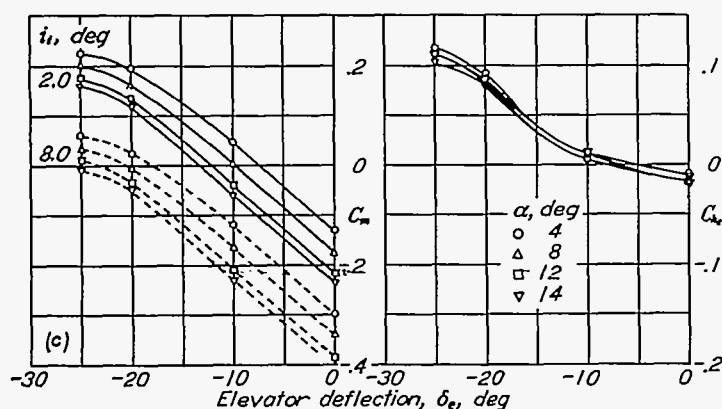


(a) Model characteristics determined from wind-tunnel tests. c.g. at 16-percent MAC.

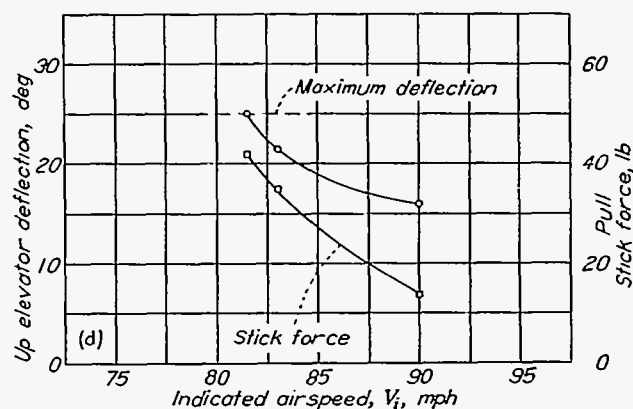
①	②	③	④	⑤	⑥	⑦	⑧	⑨	⑩	⑪	⑫	⑬	⑭	⑮	⑯	⑰
Tab neutral												Tab $C_{L_0}$ to trim at $1.4 V_{stall}$ , no ground, normal $i_t$				
$\alpha$ , angle of attack of reference line	$C_{L_W}$ in presence of ground corresponding to $\alpha$ of ①. From part (a)	$\epsilon$ , downwash angle in presence of ground corresponding to $C_{L_W}$ of ② and $\alpha$ of ①. Obtained from ref. 4	Tail off $\alpha$ with no ground effect corresponding to $C_{L_W}$ of ②. From part (a)	$\epsilon$ , downwash angle with no ground effect corresponding to $C_{L_W}$ of ② and $\alpha$ of ①. Obtained from ref. 4	$\Delta\epsilon$ , increment of downwash angle at tail due to ground effect, equals $\epsilon$ of ⑤ minus $\epsilon$ of ④	$\Delta\alpha$ due to ground effect equals ⑥ minus ⑤ as obtained from ref. 4	$\Delta\alpha_{tr}$ , the total change in tail angle of attack resulting from ground effect equals $\Delta\alpha$ of ⑥ minus $\Delta\epsilon$ of ④	$\delta_e$ for $C_m=0$ , with tail incidence increased by increment equal to $\Delta\alpha_{tr}$ of ⑦ and with angle of attack equal to $\alpha$ of ①. Obtained from part (c)	$C_L$ corresponding to $\alpha$ of ① from trim lift curve of part (a)	VI for $C_L$ of ⑨. Wing loading = 25 lb/sq ft.	$C_{L_0}$ corresponding to $\alpha$ of ①, $\Delta\alpha_{tr}$ of ⑦, and $\delta_e$ of ⑧. From part (c)	$\delta_e$ for $C_m=0$ , no ground, at $1.4 V_{stall}$ from part (b). ( $C_{L_0}=9$ , $\alpha_{trim}=4.9^\circ$ from part (a))	$C_{L_0}$ for $\delta_e$ and $\alpha_{trim}$ of ⑩, from part (c)	Tab $C_{L_0}$ equals $-C_{L_0}$ of ⑩	$C_{L_0}$ in landing equals ⑮ + ⑯	Stick force from $F/g C_{L_0} = 23.8$
11°	1.59	3.3°	12.9°	11.2°	-7.9°	-1.9°	6.0°	-25°	1.48	81.4 mph	0.105	-8.0°	0.005	-0.005	-0.100	41.4 lb pull

The above computations are for the three point attitude. Computations for landing at greater speeds are made by interpolating between the  $i_t$  limits of part (a). The results of reference 5 indicate a smaller increase in  $C_{L_W}$  at a constant attitude due to ground effect, than that computed by reference 4. Reference 5 also indicates a pitching moment increment on the wing, due to ground effect that tends to stall the airplane. The computations above do not allow for the ground effects noted in reference 5. This procedure results in a conservative estimate of  $\delta_e$  and stick force to land (with respect to reference 5).

(b) Computation table.

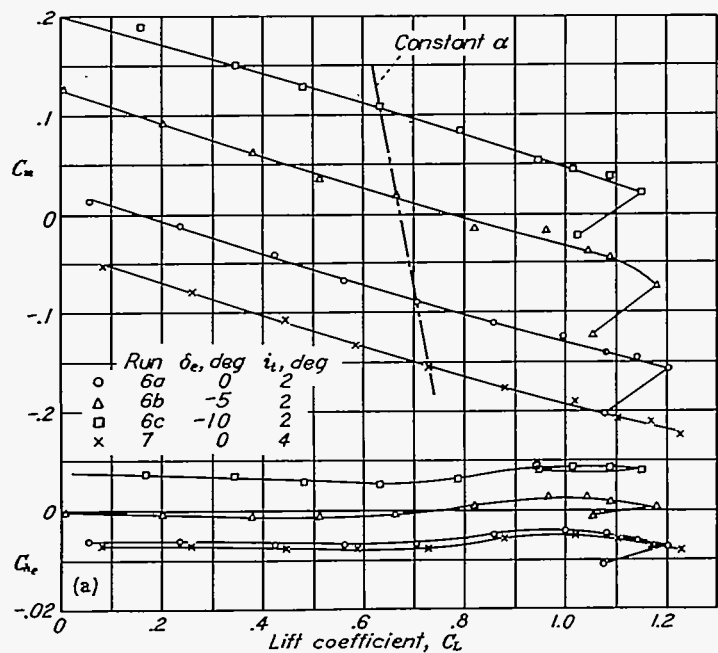


(c) Model characteristics cross-plotted.



(d) Airplane landing characteristics.

FIGURE 3.—Variation of elevator angle and stick force in landing. Flaps and gear down, propeller windmilling. Single-engine airplane.



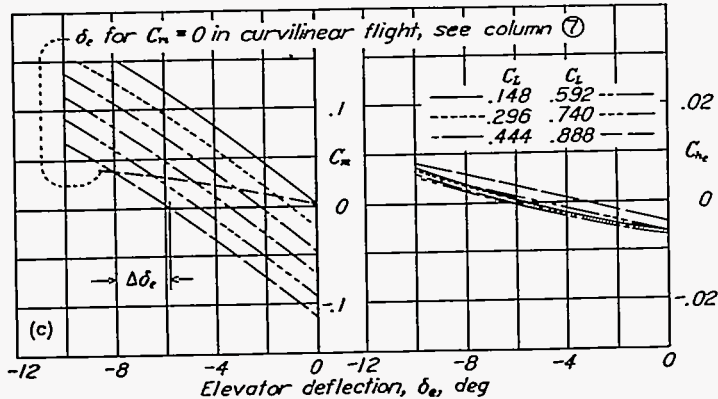
(a) Model characteristics determined from wind-tunnel tests. a. g. at 26-percent MAC.

①	②	③	④	⑤	⑥	⑦	⑧	⑨	⑩	⑪	⑫	⑬
n	C <sub>L</sub> for values of ②	δ <sub>e</sub> to balance C <sub>m</sub> due to static stability. From crossplot	$\frac{n^2-1}{n}$	Δα <sub>i</sub> due to curvilinear flight	Δδ <sub>e</sub> to balance C <sub>m</sub> due to Δα <sub>i</sub>	Neutral tab				Tab C <sub>L</sub> with tab set for trim at n of 1	C <sub>L</sub> in curvilinear flight with trim at n of 1 equals ⑫+⑬	Stick force from $\frac{F}{gC_{L_0}}=23.8$
						δ <sub>e</sub> for C <sub>m</sub> =0 in curvilinear flight equals ⑦ plus ⑥	C <sub>L</sub> for ⑧ and ⑨ from crossplot	C <sub>L</sub> due to Δα <sub>i</sub>	C <sub>L</sub> in curvilinear flight equals ⑩ plus ⑪			
1	0.148	0.1°	0	0°	0°	0.1°	-0.0062	0	-0.0062	+0.0062	0	0
2	.296	-1.1°	3/2	0.31°	-0.54°	-1.6°	-0.0048	-0.0002	-0.0050	+0.0062	0.0012	6.3
3	.444	-2.2°	8/3	.55°	-0.96°	-3.2°	-0.0034	-0.0003	-0.0037	+0.0062	.0025	13.1
4	.592	-3.3°	15/4	.77°	-1.34°	-4.9°	-0.0018	-0.0004	-0.0022	+0.0062	.0040	21.0
5	.740	-4.8°	24/5	1.03°	-1.79°	-6.8°	+0.0018	-0.0005	+0.0013	+0.0062	.0075	39.4
6	.888	-5.9°	35/6	1.20°	-2.09°	-8.0°	+0.0055	-0.0006	+0.0049	+0.0062	.0111	58.0

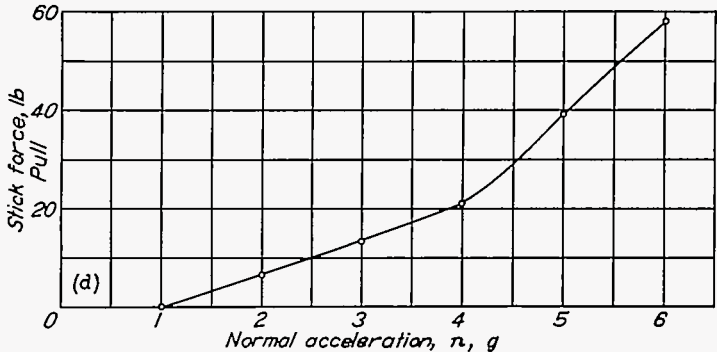
$$\Delta\alpha_i = \frac{l_H g}{(V_{TP})^2} \left( \frac{n^2-1}{n} \right) 57.3 = 0.206 \left( \frac{n^2-1}{n} \right) \quad \textcircled{7} = \Delta\alpha_i \left( \frac{dC_{L_0}}{d\alpha_i} \right)_{\alpha} = -0.0005\Delta\alpha_i$$
$$\Delta\delta_e = \Delta\alpha_i / \left( \frac{\partial\alpha_i}{\partial\delta_e} \right) C_{L_i} = -\Delta\alpha_i / \left( \frac{dC_m}{d\delta_e} \right)_{\alpha} = -\frac{\Delta\alpha_i}{0.575}$$

V<sub>T</sub> = 294 m. p. h. at 9,000 ft., wing loading = 32.6 lb/sq ft.

(b) Computation table.



(c) Model characteristics cross-plotted.



(d) Airplane accelerated flight stick-force characteristics.

FIGURE 4. Variation of elevator stick force with normal acceleration in steady turning flight. Flaps and gear up, propeller windmilling. Single-engine airplane.



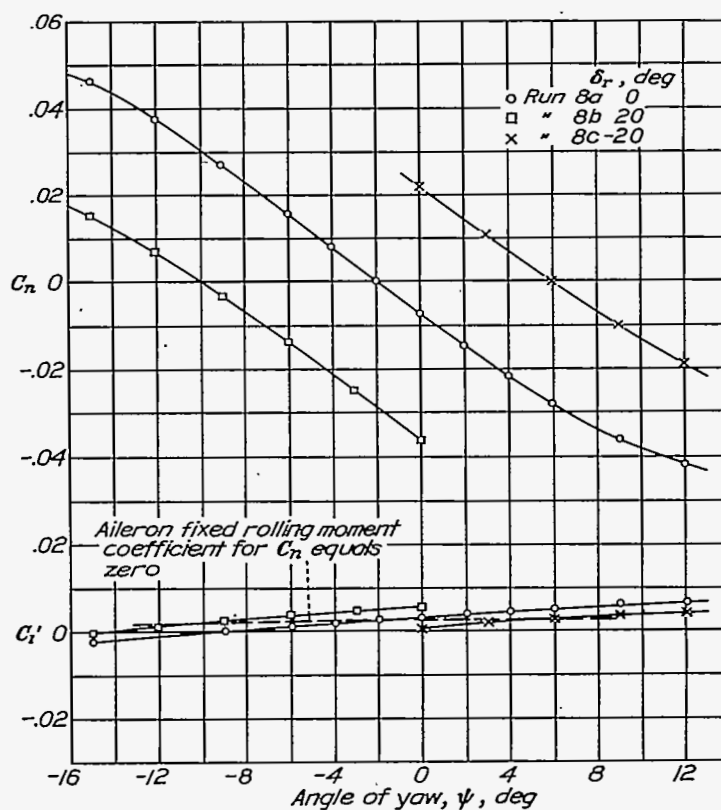
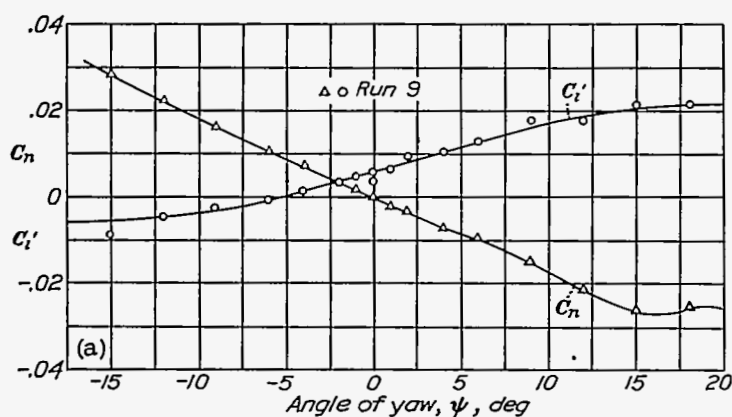
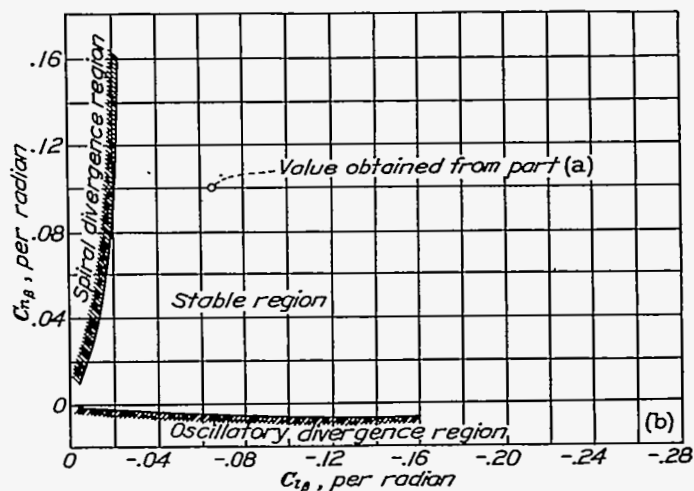


FIGURE 5.—Dihedral characteristics at low speed. Flaps and gear down, 50-percent normal rated power. Single-engine airplane.

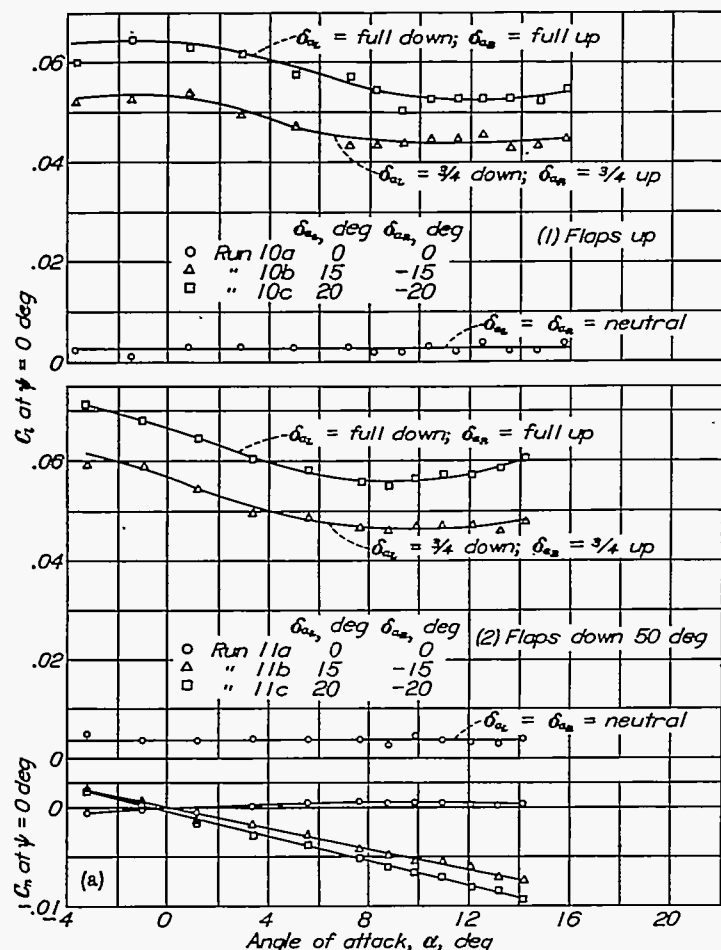


(a) Model characteristics determined from wind-tunnel tests.



(b) Divergence characteristics of airplane.

FIGURE 6.—Lateral-stability characteristics at high speed. Flaps and gear up, rated power. Single-engine airplane.



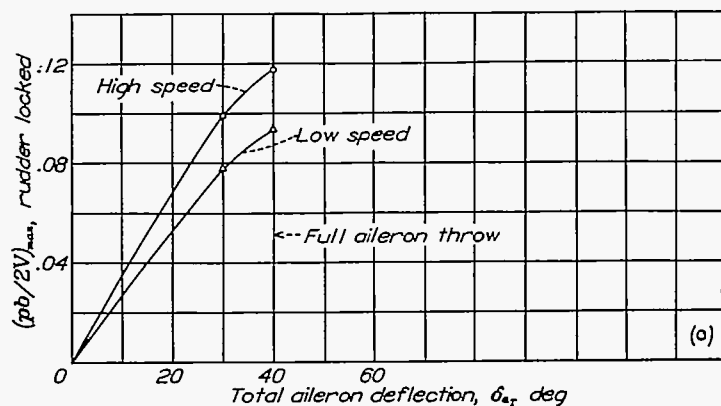
(a) Model characteristics determined from wind-tunnel tests.

FIGURE 7.—Aileron control characteristics ( $pb/2V$  against aileron deflection). Flaps and gear up. Single-engine airplane.

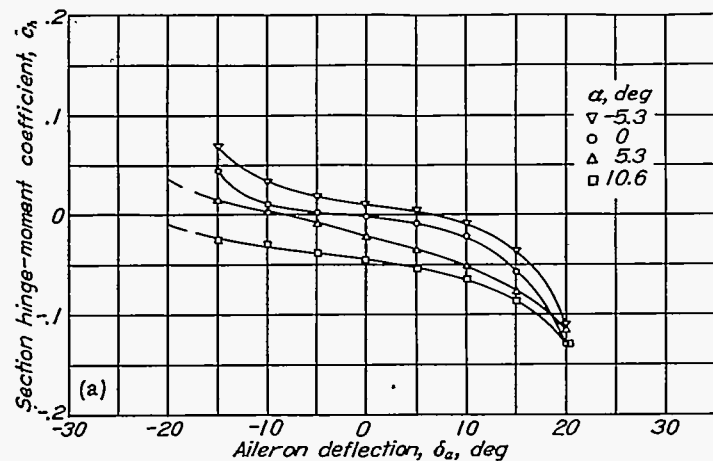
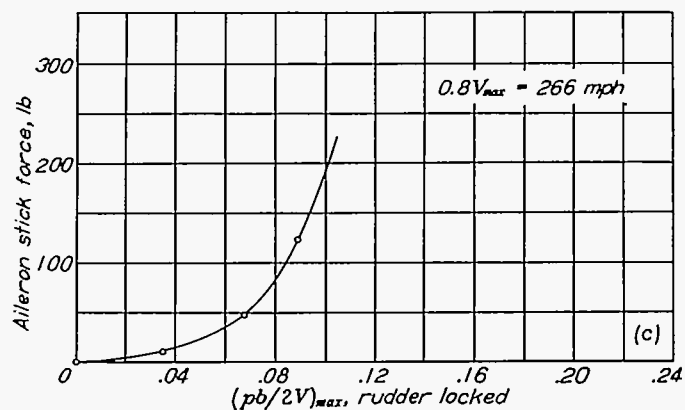
Condition	Aileron throw	Rolling moment due to ailerons. From fig. (a-1)	zero sideslip, equals $\frac{C_{L\beta}}{C_{L\alpha}} = \frac{C_{L\beta}}{C_{L\alpha}}$	reduction factor—due to sideslip $\frac{C_{L\beta}}{C_{L\alpha}}$	rudder-locked; $\frac{C_{L\beta}}{C_{L\alpha}}$	Total aileron deflection, $\delta_{ar}$
High-speed <sup>1</sup>	$\frac{3}{4}$	0.051	0.109	0.91	0.099	30°
	Full	.061	.130	.91	.118	40°
Low-speed <sup>2</sup>	$\frac{3}{4}$	.042	.097	.80	.078	30°
	Full	.060	.116	.80	.093	40°

<sup>1</sup>  $V_1 = 0.8 V_{max} = 266$  mph;  $C_L = 0.18$ ,  $\alpha = 0.0^\circ$ .<sup>2</sup>  $1.2 V_{stall}$ ;  $C_L = 0.88$ ,  $\alpha = 10.4^\circ$ .<sup>3</sup>  $C_{L\beta} = -0.47$  at high speed and  $-0.43$  at low speed (from ref. 6).<sup>4</sup> Believed to be representative of modern high performance airplanes. (Assumes a rigid wing.)

(b) Computation table, flaps up.



(c) Airplane rolling characteristics with flaps up.

(a) Hinge-moment characteristics determined from wind-tunnel tests.  
Note: Hinge-moment coefficients obtained from two-dimensional data.

(c) Aileron stick-force characteristics in steady rolls.

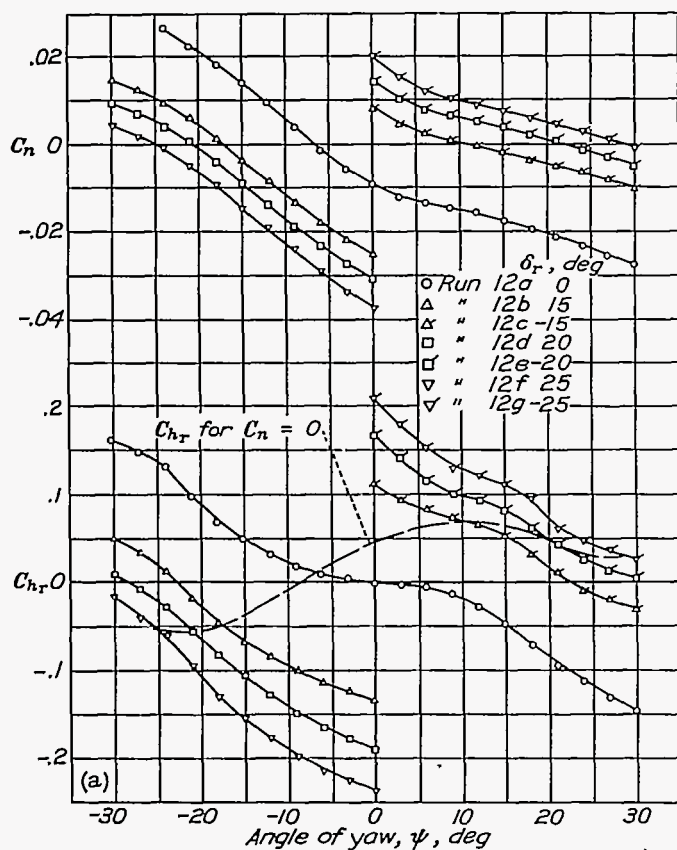
Aileron position	Left aileron deflection, $\delta_{aL}$	Right aileron deflection, $\delta_{aR}$	$\left(\frac{pb}{2V}\right)_{max}$ from fig. 7	Induced angle due to rolling, $\Delta\alpha = 40 \times \frac{1}{10}$	Average angle of attack over each aileron <sup>1</sup>	$C_{h_{aL}}$ for $\delta_{aL}$ and $\alpha_L$ from (a) above	$C_{h_{aR}}$ for $\delta_{aR}$ and $\alpha_R$ from (a) above	Summation $C_{h_a}$ ⑧-⑦	Stick force $\frac{f}{q C_{h_a}}$ ⑩	Aileron control force in lb $f = ⑩ \times q \times ⑪$
0	0	0	0	0	0	-0.002	-0.002	0	12.3	0
$\frac{1}{4}$ throw	5°	-5°	0.035	1.4°	-1.4°	-0.005	0	-0.005	12.3	11
$\frac{1}{2}$ throw	10°	-10°	.068	2.7°	-2.7°	-0.015	.006	-.021	12.3	47
$\frac{3}{4}$ throw	15°	-15°	.089	3.6°	-3.6°	-.044	.021	-.065	12.3	123
Full throw	20°	-20°	.118	4.7°	-4.7°	-.118	.041	-.159	12.3	354

<sup>1</sup>  $\Delta\alpha = 40 \times \left(\frac{pb}{2V}\right)_{max} = \frac{l_1 + l_2}{b} \times \left(\frac{pb}{2V}\right) \times 57.3$  where  $l_1$  and  $l_2$  are distances from plane of symmetry to inboard and outboard ends of the aileron.<sup>2</sup>  $\alpha = 0.0^\circ$  at  $0.8 V_{max} = 266$  mph;  $q = 181$  lb/sq ft.

(b) Computation table.

FIGURE 8.—Variation of aileron stick force with  $pb/2V$  at high speed. Flaps and gear up. Single-engine airplane.





①	②	③	④	⑤	⑥
Rudder angle, $\delta_r$	$\psi$ for $C_n$ equals zero	$\beta$ for $C_n=0$ equals $-\psi$ for $C_n=0$	$C_{h_r}$ for ① and ②	Pedal force $\frac{qC_{h_r}}{g}$	Pedal force, lb
-25° R	28.5°	28.5° L	0.030	28.6	25 R
-20° R	21.5°	21.5° L	0.040	28.6	34 R
-15° R	10.5°	10.5° L	0.070	28.6	59 R
0°	0.7°	0.7° R	0.010	28.6	8 R
15° L	-17.2°	17.2° R	-0.050	28.6	42 L
20° L	-20.7°	20.7° R	-0.057	28.6	48 L
25° L	-25.0°	25.0° R	-0.055	28.6	46 L

$V_t=111$  mph  $q=31.5$  lb/sq ft., wing loading= $32.6$  lb/sq ft.

(b) Computation table.

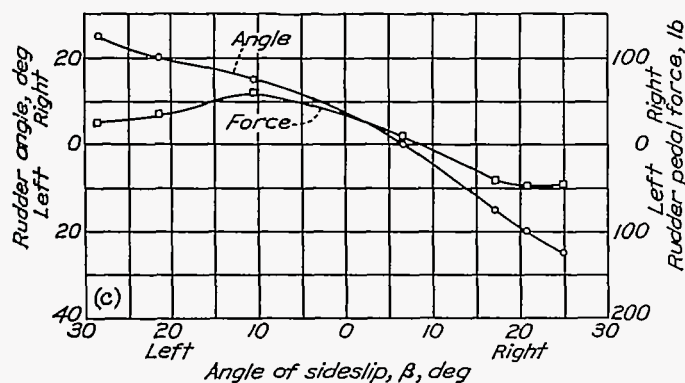
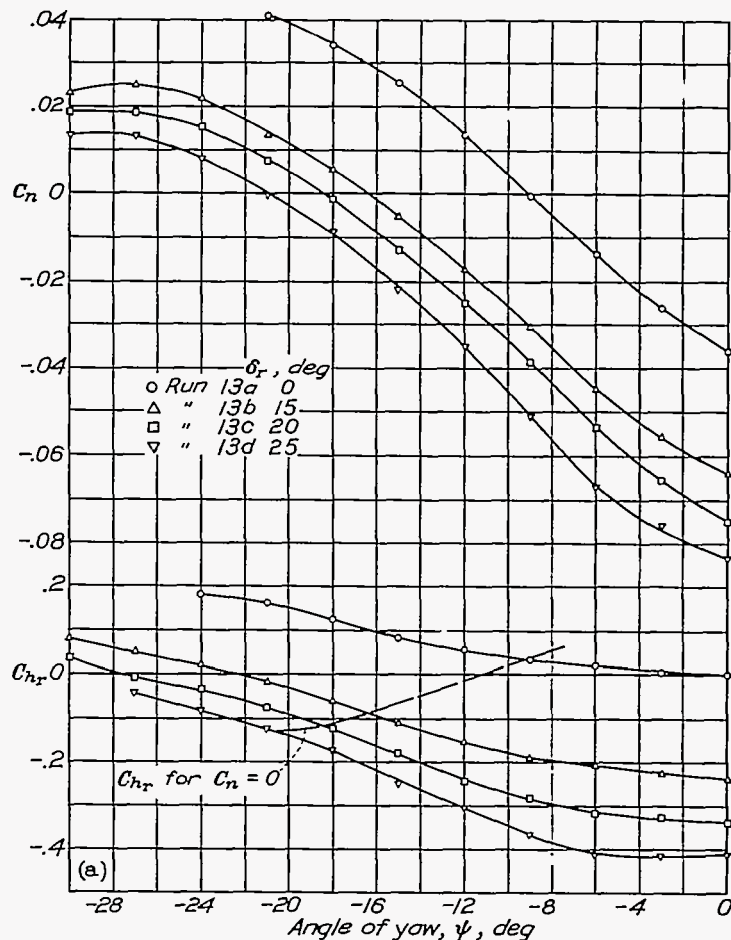


FIGURE 9.—Variation of rudder angle and pedal force with sideslip at low speed. Flaps and gear up, normal rated power. Single-engine airplane.



①	②	③	④	⑤	⑥
Rudder angle, $\delta_r$	$\psi$ for $C_n$ equals zero	$\beta$ for $C_n=0$ equals $-\psi$ for $C_n=0$	$C_{h_r}$ for ① and ②	Pedal force $\frac{qC_{h_r}}{g}$	Pedal force, lb
0	-9.1°	9.1° R	0.036	28.6	16 R
15° L	-16.4°	16.4° R	-0.090	28.6	40 L
20° L	-18.4°	18.4° R	-0.119	28.6	53 L
25° L	-20.9°	20.9° R	-0.130	28.6	58 L

$V_t=81$  mph,  $q=16.8$  lb/sq ft., wing loading= $32.6$  lb/sq ft.

(b) Computation table.

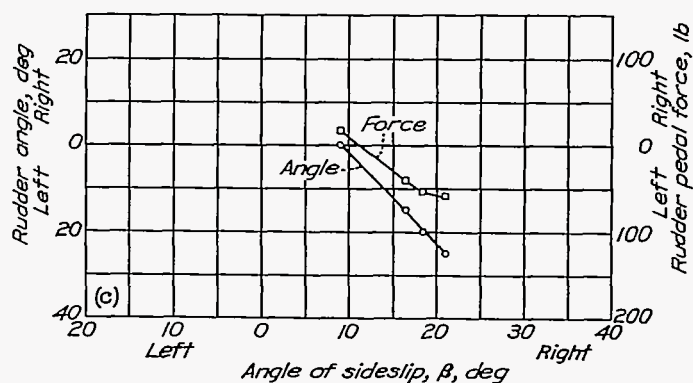
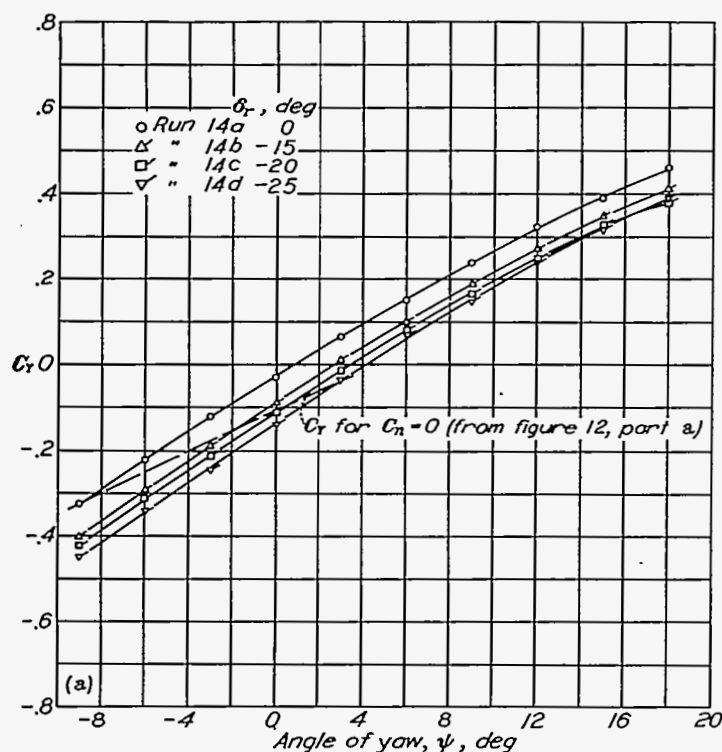


FIGURE 10.—Variation of rudder angle and pedal force with sideslip in wave-off. Flaps and gear down, take-off power. Single-engine airplane.

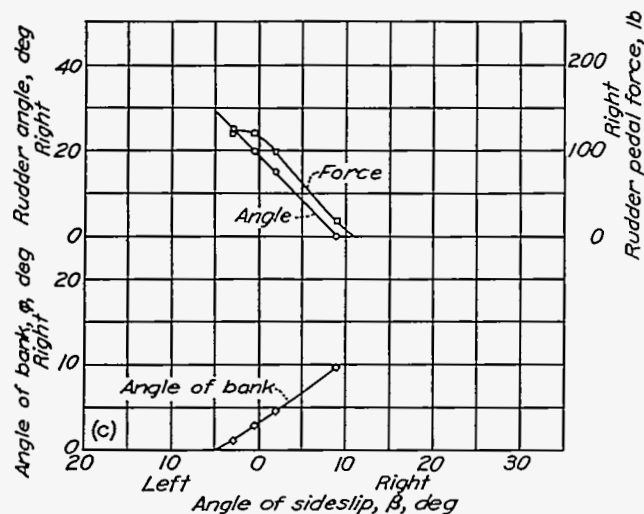


(a) Model characteristics determined from wind-tunnel tests.

①	②	③	④	⑤	⑥	⑦	⑧
Rudder angle, $\delta_r$	$\psi$ for $C_n$ equals zero	$\beta$ for $C_n=0$ equals $-\psi$ for $C_n=0$	$C_{\delta_r}$ for ① and ②	Pedal force $\frac{q C_{\delta_r}}{q C_{\delta_r}}$	Pedal force, lb	$C_r$ for ① and ②	Angle of bank $\phi$ equals $\sin^{-1} \frac{C_r}{C_L}$
Data taken from fig. 12, pt. (a)						From pt. (a) fig. 11	
0°	-9.0°	9.0° R	+0.040	26.6	17 R	-0.325	9.6° R
-15° R	-1.9°	1.9° R	.222	26.6	98 R	-.150	4.4° R
-20° R	+0.5°	0.5° L	.271	26.6	131 R	-.096	2.8° R
-25° R	2.9°	2.9° L	.372	26.6	121 R	-.040	1.2° R

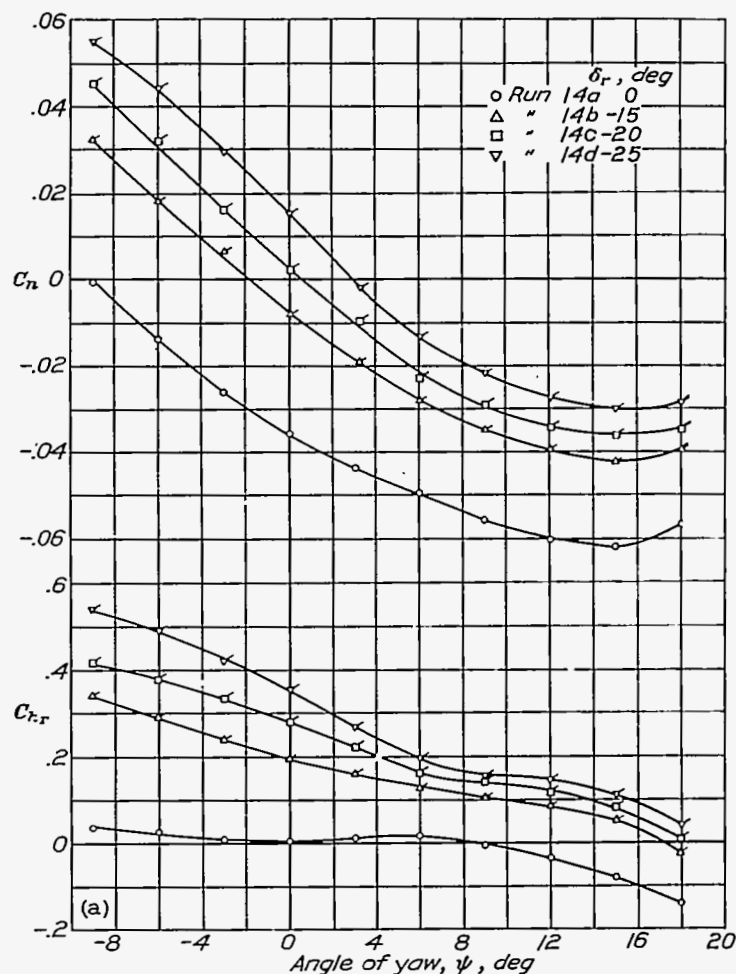
$V_i=81$  mph,  $q=16.8$  lb/sq ft., wing loading = 32.6 lb/sq ft.

(b) Computation table.



(c) Airplane steady sideslip characteristics.

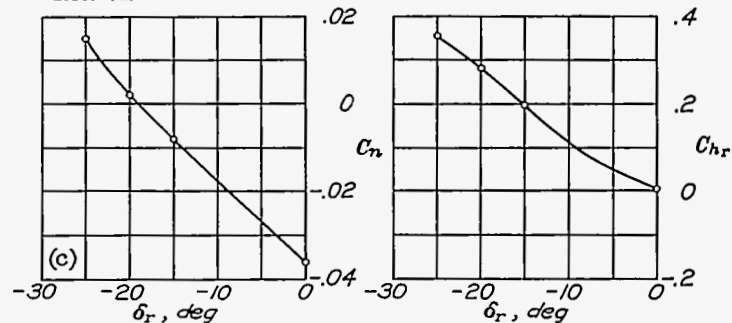
FIGURE 11.—Rudder angle and pedal force necessary to hold wings level in wave-off. Flaps and gear down, take-off power. Single-engine airplane.



(a) Model characteristics determined from wind-tunnel tests.

①	②	③	④	⑤	⑥	⑦
$C_L$ for full right alleron, see fig. 7 $\alpha=6^\circ$	$C_n$ for full right alleron, see fig. 7 $\alpha=6^\circ$	$C_{\delta_r}$ from ref. 6	$\frac{p b^2 V}{q S b}$ equals $-\frac{C_n}{C_L}$	$C_{\delta_r}$ from ref. 6	$C_n$ due to rolling equals ① × ⑤	$C_n$ to be overcome by rudder = ⑥ + ⑦
0.053	-0.0050	-.43	0.123	-0.017	-0.0050	-0.0109

(b) Computation table. Yawing-moment coefficient due to maximum right alleron deflection.

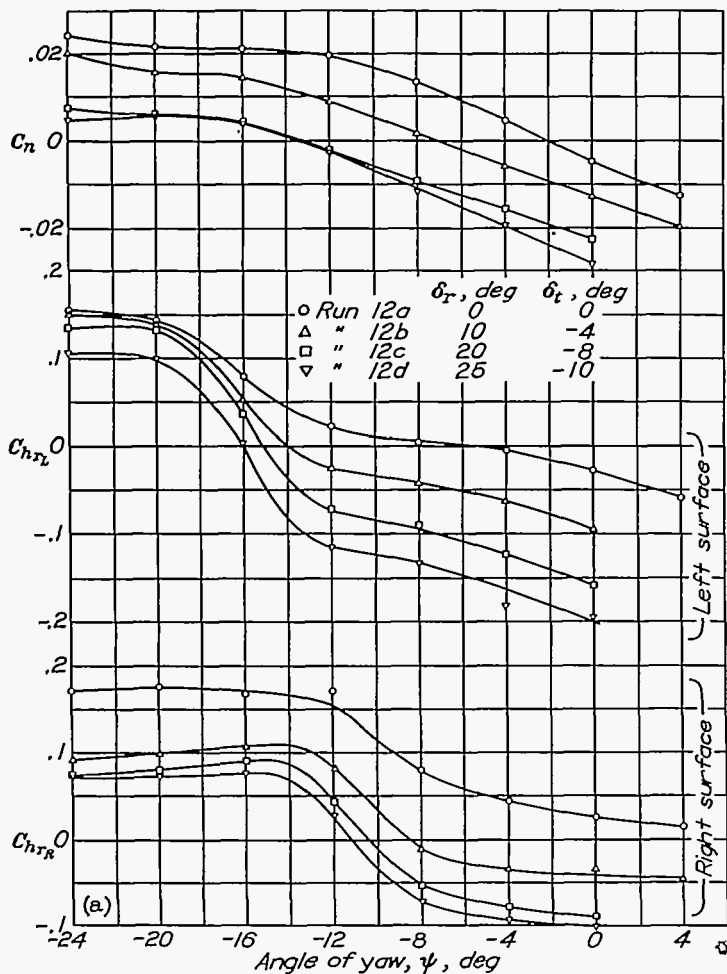
(c) Model characteristics of part (a) cross-plotted at  $\psi=0^\circ$ 

①	②	③	④	⑤
$C_n$ to be produced by rudder, from ② of (b) above.	$\delta_r$ corresponding to ① from crossplot (c) above.	$C_{\delta_r}$ corresponding to ② from crossplot (c) above.	Pedal force $\frac{q C_{\delta_r}}{q C_{\delta_r}}$	Pedal force, lb
0.0109	-23.5° R	0.335	26.6	160 R

$V_i=81$  mph,  $q=16.8$  lb/sq ft., wing loading = 32.6 lb/sq ft.

(d) Computation table. Rudder angle and pedal force to hold zero sideslip with maximum right alleron.

FIGURE 12.—Rudder angle and pedal force necessary to hold zero sideslip in wave-off. Flaps and gear down, take-off power. Single-engine airplane.



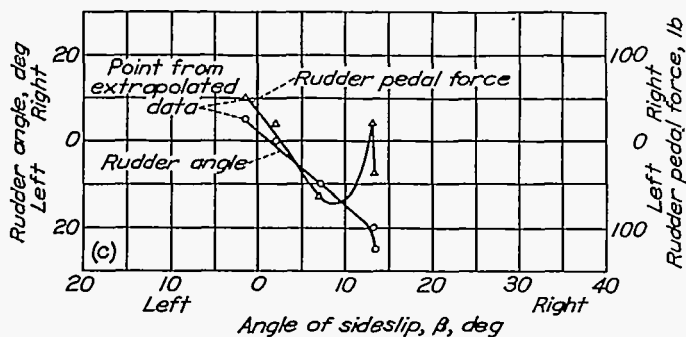
(a) Model characteristics determined from wind-tunnel tests.

Note: Each rudder is equipped with a boost tab, which is deflected for the runs.

①	②	③	④		⑤	⑥	⑦
Rudder angle, $\delta_r$	$\psi$ for $C_n=0$	$\beta$ for $C_n=0$ equals $-\psi$ for $C_n=0$	$C_{nr}$ for ① and ②		$C_{nr}$ total, ⑤+⑥	Pedal force $gC_{nr}$	Pedal force, lb
			Left $C_{nrL}$	Right $C_{nrR}$			
0	-2.0°	2.0° R	-0.014	+0.034	+0.020	38.5	20 R
10°	-7.1°	7.1° R	-.014	-.021	-.065	38.5	64 L
20°	-13.2°	13.2° R	-.055	+.075	+.020	38.5	20 R
25°	-13.4°	13.4° R	-.097	.000	-.037	38.5	37 L

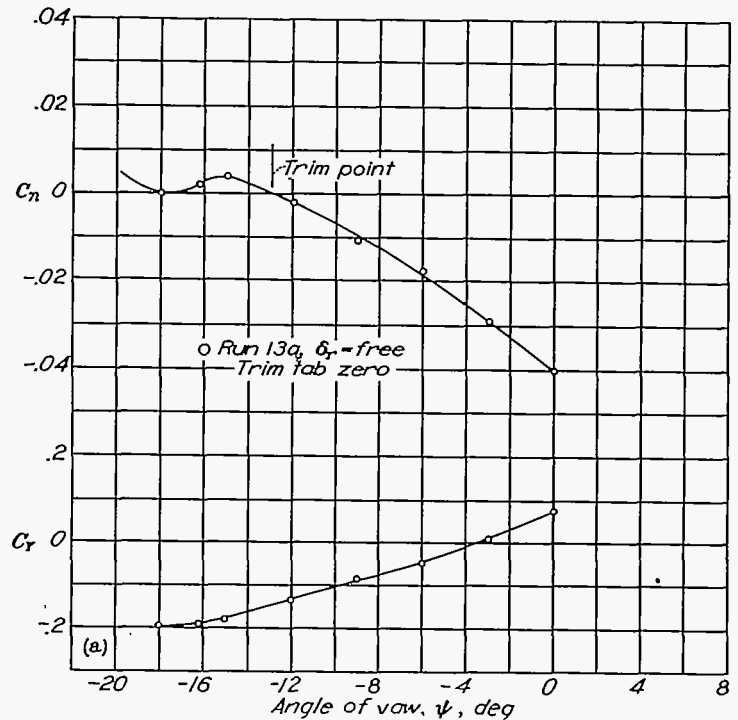
$V_1=100$  mph,  $q=25.6$  lb/sq ft., wing loading=48 lb/sq ft.

(b) Computation table.



(c) Airplane steady sideslip characteristics.

FIGURE 13.—Variation of rudder angle and pedal force with sideslip at approach speed. Flaps and gear down, take-off power. Twin-engine, twin-tail airplane.

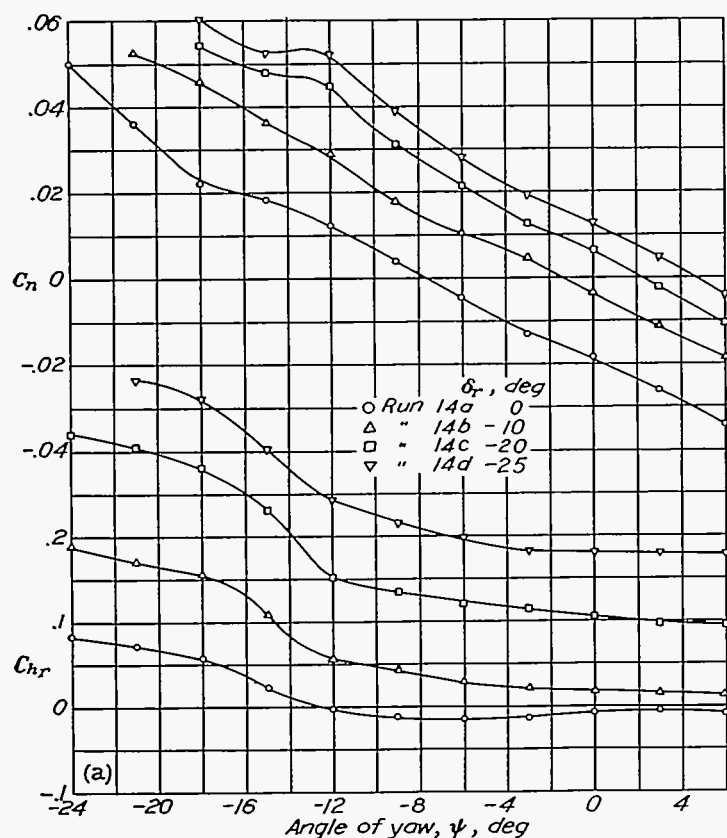


(a) Model characteristics determined from wind-tunnel tests.

①	②	③	④
Angle of yaw at which $C_n$ equals zero	Side force coefficient, $C_r$ , at ①	Airplane lift coefficient $C_L = \frac{W}{qS}$	Angle of bank, $\varphi = \sin^{-1} \frac{C_r}{C_L}$
-12.9°	-0.15	1.70	5.1° right

(b) Angle of bank computation table.

FIGURE 14.—Rudder-free trim characteristics with asymmetric power at low speeds. Flaps and gear down, take-off power on right engine. Left engine propeller windmilling. Twin-engine airplane.



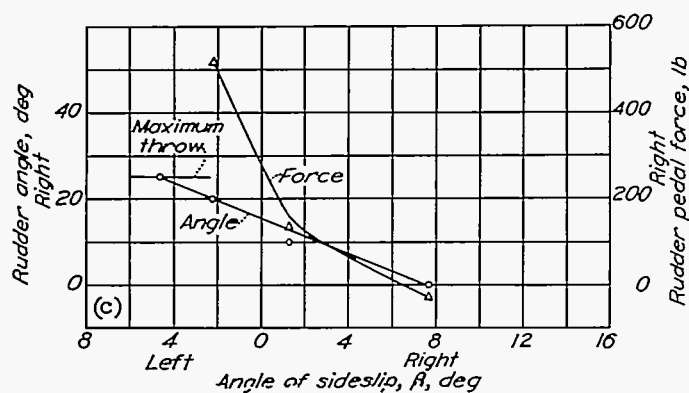
(a) Model characteristics determined from wind-tunnel tests.

①	②	③	④	⑤	⑥	⑦	⑧
Rudder angle, $\delta_r$	$\psi$ for $C_n=0$	$\beta$ for $C_n=0$ equals $-\psi$ for $C_n=0$	$C_{L_r}$ for ① and ②	$\Delta C_{L_r}$ of trim tab set to trim with symmetric power <sup>1</sup>	$C_{L_r}$ total ⑤+⑥	Pedal force $qC_{L_r}$	Pedal force, lb
0°	-7.7°	7.7° R	-0.014	+0.008	-0.006	110.6	29 L
-10°	-1.3°	1.3° R	+0.020	+0.008	+0.028	110.6	133 R
-20°	+2.2°	2.2° L	.100	+0.008	.108	110.6	515 R
-25°	4.6°	4.6° L	.180	+0.008	.188	110.6	896 R

$V_i=130$  mph,  $q=43.1$  lb/sq ft., wing loading=45 lb/sq ft.

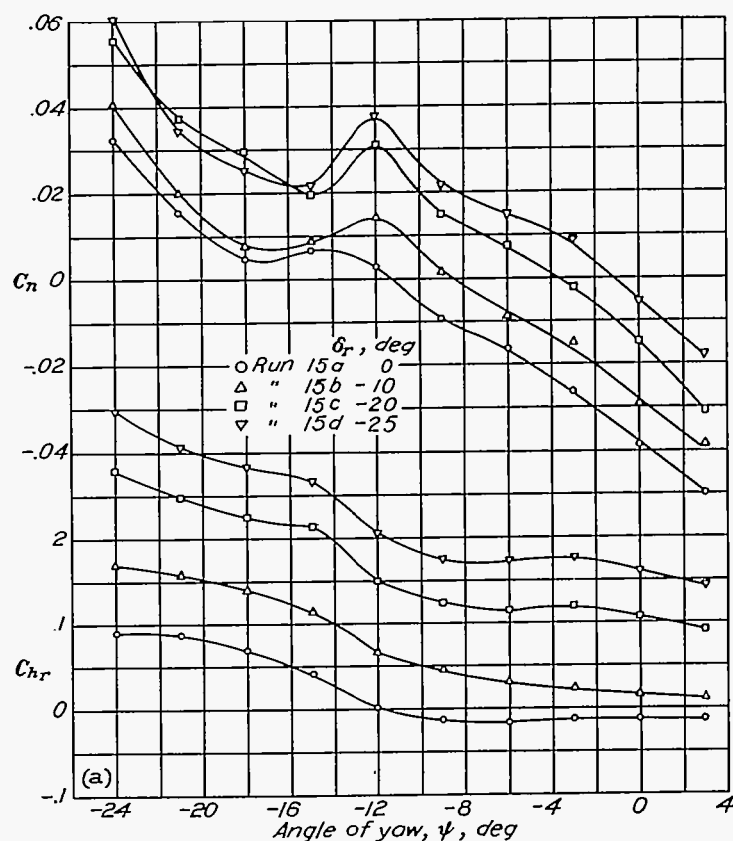
<sup>1</sup> Estimated from Item VIIIa.

(b) Computation table.



(c) Airplane steady sideslip characteristics.

FIGURE 15.—Rudder angle and pedal force necessary to hold zero sideslip with asymmetric power at low speed. Flaps and gear up, take-off power on right engine, left engine propeller windmilling. Twin-engine airplane.



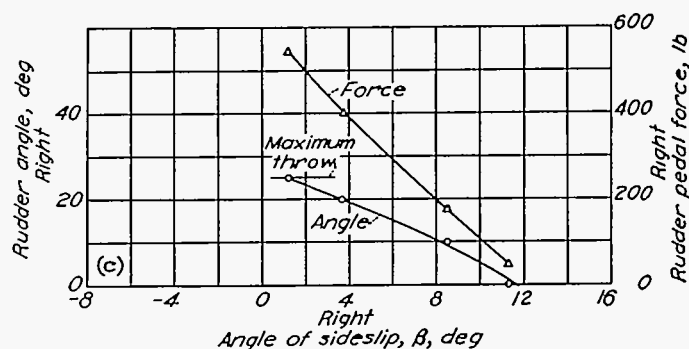
(a) Model characteristics determined from wind-tunnel tests.

①	②	③	④	⑤	⑥	⑦	⑧
Rudder angle, $\delta_r$	$\psi$ for $C_n=0$	$\beta$ for $C_n=0$ equals $-\psi$ for $C_n=0$	$C_{L_r}$ for ① and ②	$\Delta C_{L_r}$ of trim tab set to trim with symmetric power <sup>1</sup>	$C_{L_r}$ total ⑤+⑥	Pedal force $qC_{L_r}$	Pedal force, lb.
0	-11.3°	11.3° R	-0.003	0.018	0.015	110.6	44 R
-10°	-8.5°	8.5° R	+0.041	.018	.059	110.6	173 R
-20°	-3.7°	3.7° R	.118	.018	.136	110.6	400 R
-25°	-1.2°	1.2° R	.168	.018	.186	110.6	540 R

$V_i=102$  mph,  $q=26.6$  lb/sq ft., wing loading=45 lb/sq ft.

<sup>1</sup> Estimated from Item VIIIa.

(b) Computation table.



(c) Airplane steady sideslip characteristics.

FIGURE 16.—Rudder angle and pedal force necessary to hold 10° sideslip with asymmetric power at low speed. Flaps and gear down, take-off power on right engine, left engine propeller windmilling. Twin-engine airplane.

TABLE I.—LONGITUDINAL CHARACTERISTICS

[Single- and twin-engine airplanes]

Item	Purpose and requirement	Critical condition	Run No.	Description of run	Fig. No.	Remarks
I	To determine if the horizontal tail is large enough to meet requirement of stick-fixed stability and if the elevator-floating characteristics are such as to maintain stick-free stability under the specified flight condition.	Critical condition will be in a rated-power climb, where destabilizing effects of power at normal flight speeds will be maximum. Speed range over which stability is required is to be determined from particular specification being followed. or Critical condition may occur in the approach, with flaps and gear down and 50-percent normal rated power.	1a----- 1b-----	Polars with rated power, flaps and gear up. $\delta_e = 0^\circ$ $\delta_e = -5^\circ$	1	
			2a----- 2b----- 2c-----	Polars with 50-percent normal rated power, flaps and gear down. $\delta_e = 0^\circ$ $\delta_e = -5^\circ$ $\delta_e = -10^\circ$	2	
II	To determine if elevator is large enough for necessary control under all normal flight conditions.	Critical condition will be in landing where forwardmost center-of-gravity location and ground effect will require maximum up-elevator to secure landing attitude.	3a----- 3b----- 3c----- 3d-----	Polars with propeller windmilling, flaps and gear down. $\delta_e = 0^\circ$ $\delta_e = -10^\circ$ $\delta_e = -20^\circ$ $\delta_e = -25^\circ$ $i_t = \text{normal} + \Delta\alpha_t$	3	Runs (4a, b, c) with increased incidence are for the purpose of determining $dC_m/di$ and $dC_{L_0}/di$ , necessary for application of method of reference 4. $\Delta\alpha_t$ should be selected as change in angle of attack of the tail in the minimum speed landing attitude, computed by method of reference 4.  Run 5, with tail removed, required for determination of $C_{L_{\alpha}}$ used to compute $\Delta\alpha$ and $\Delta\alpha_t$ by method of reference 4.
			4a----- 4b----- 4c----- 4d----- 5-----	$\delta_e = 0^\circ$ $\delta_e = -10^\circ$ $\delta_e = -20^\circ$ $\delta_e = -25^\circ$ Tail off		
III	To determine if elevator balance is sufficient to maintain control forces within required limits.	Critical condition will either be in landing or in accelerated flight with propellers windmilling where stability will be greatest and consequently the stick force per $g$ the highest. In landing, a maximum force of 35 pounds for stick-type control and 50 pounds for wheel-type control is permissible (with trim tab set at $1.4V_{stall}$ , propeller windmilling). The required stick-force gradient in accelerated flight varies with type of airplane and must be determined from flying-qualities specifications being followed.	6a----- 6b----- 6c----- 7-----	Landing: Data required are same as for II above. Accelerated Flight: Polars with propellers windmilling, flaps and gear up.  $\delta_e = 0^\circ$ $\delta_e = -5^\circ$ $\delta_e = -10^\circ$ $i_t = \text{normal} + \Delta\alpha_t$ $\delta_e = 0^\circ$	4	Run 7, with increased incidence, made for the purpose of determining $dC_m/di$ and $dC_{L_0}/di$ , necessary for accelerated flight calculations. Value of $\Delta\alpha_t$ should be determined as maximum induced angle at tail in accelerated flight.

TABLE II.—LATERAL CHARACTERISTICS

[Single- and twin-engine airplanes]

Item	Purpose and requirement	Critical condition	Run No.	Description of run	Fig. No.	Remarks
IV	To determine if the wing dihedral is great enough to provide at least neutral dihedral effect for the conditions of flight specified.	The critical condition will be in the approach with flaps down and with power on where power and flap effects combine to reduce the dihedral effect. (This condition will normally be worse with ailerons free, but it can be checked to a very good first approximation with ailerons fixed.)	8a----- 8b----- 8c-----	Yaw run at approach attitude with flaps and gear down and 50-percent normal rated power. $\delta_e = 0^\circ, \psi = -30^\circ \text{ to } 30^\circ$ $\delta_e = 20^\circ, \psi = -30^\circ \text{ to } 0^\circ$ $\delta_e = -20^\circ, \psi = 0^\circ \text{ to } 30^\circ$	5	Army calls for stability at $1.2V_{stall}$ (propeller windmilling) with 50-percent rated power. Navy calls for stability in "the approach with considerable power." This condition will normally coincide with the condition outlined for the Army above. The angle of attack for these tests should be chosen on the basis of $C_{L_{max}}$ obtained in the wind tunnel (used in computing $1.2V_{stall}$ ) but the power ( $T_e$ ) should be set in accordance with the estimated speed under full-scale conditions.
V	To determine if proper balance exists between dihedral effect and directional stability to avoid oscillatory divergence.	Critical condition will be the high-speed (clean) condition where dihedral effect will be maximum and directional stability minimum (due to small power effects).	9-----	Yaw run at high-speed attitude, flaps and gear up, propeller windmilling (or highspeed $T_e$ ). $\delta_e = 0^\circ, \psi = -30^\circ \text{ to } 30^\circ$	6	Some doubt exists as to whether or not this criterion expresses a true maximum limit for dihedral. It is believed that an airplane can have dihedral under this limit and yet have an undesirably large roll due to sideslip, and that the tolerable amount actually varies with the type of airplane. However no specific requirement expressing such a criterion exists.
VI	To determine if ailerons are sufficiently effective to furnish minimum $(\frac{pb}{2V})_{max}$ required.	Critical condition will be at low speed (flaps up or down) where aileron effectiveness is usually lowest and reduction in $(\frac{pb}{2V})_{max}$ due to yawing is greatest.	10a----- 10b----- 10c-----  11a----- 11b----- 11c-----	Polar with windmilling propeller. Flaps and gear retracted. $\delta_{eL} = 0, \delta_{eR} = 0$ $\delta_{eL} = \frac{3}{4}$ Down, $\delta_{eR} = \frac{3}{4}$ Up $\delta_{eL} = \text{Full Down}, \delta_{eR} = \text{Full Up}$  Flaps and gear extended. $\delta_{eL} = 0, \delta_{eR} = 0$ $\delta_{eL} = \frac{3}{4}$ Down, $\delta_{eR} = \frac{3}{4}$ Up $\delta_{eL} = \text{Full Down}, \delta_{eR} = \text{Full Up}$	7	For a single-engine airplane runs 11a, b, and c are needed for computations of necessary rudder balance. See Table No. III.
VII	To determine if ailerons are closely enough balanced to furnish required $(\frac{pb}{2V})_{max}$ with low enough control forces.	Critical condition will be at highest speed at which requirement applies, normally $0.8V_{max}$ . Required force and rate of roll varies with type of airplane.		Data required will be furnished by runs 10a, b, and c, supplemented by two-dimensional hinge-moment data.	8	For conventional-type ailerons there are normally sufficient two-dimensional data at high Reynolds number which will form a reliable basis for stick-force computations.



TABLE IV.—DIRECTIONAL CHARACTERISTICS

[Twin-engine airplanes]

[illegible]



## APPENDIX

### SYMBOLS

$\delta$	deflection of control surface
$\Delta\alpha$	change in angle of attack at wing due to ground effect or change in angle of attack (over aileron station) due to roll
$\Delta\epsilon_1$	change in down-wash at tail due to ground effect
$\Delta\epsilon_{t_1}$	change in angle of attack of tail due to ground effect
$\Delta\alpha_{t_2}$	change in angle of attack of tail due to induced angle in accelerated flight
$C_{LW}$	lift coefficient of wing and fuselage (exclusive of tail)
$i_t$	angle of incidence of tail
$C_h$	hinge-moment coefficient
$\psi$	angle of yaw
$\beta$	angle of sideslip
$T_c$	propeller thrust coefficient $\left(\frac{\text{Thrust}}{\rho V^2 D^2}\right)$
$C_{n\beta}$	yawing moment due to sideslip
$C_{l\beta}$	rolling moment due to sideslip
$C_{lp}$	rolling moment due to rolling
$C_{la}$	rolling moment due to aileron deflection
$F$	stick force, pounds
$V_i$	indicated airspeed
$l_H$	length from center of gravity to 25-percent M. A. C. of horizontal tail
$n$	normal acceleration
$g$	acceleration due to gravity
$p$	rolling velocity, radians per second

$b$	wing span, feet
$q$	dynamic pressure $\left(\frac{1}{2}\rho V^2\right)$ , pounds per square foot

#### Subscripts:

$e$	elevator
$r$	rudder
$a_L$	left aileron
$a_R$	right aileron
$t$	tail
$L$	left
$R$	right

NOTE.—Stability axes have been used in the presentation of the data. Positive deflection of control surface is in the direction which will produce a positive force (not necessarily a positive moment).

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